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MARINER II - AN EXAMPLE OF A STABILIZED INTERPLANETARY SPACE VEHICLE

by

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I. INTRODUCTION

It is my intention in this paper to explore the general considerations that led to the design concepts incorporated into a planetary spacecraft. The highly successful Mariner II spacecraft will be the example. A brief description of Mariner II will be presented with a summary of its Venusian flight history. In conclusion, some of the interesting engineering results are described and discussed.

author

II. MARINER II ACHIEVEMENTS

The value of an attitude stabilized spacecraft for the exploration of space has been demonstrated by the Mariner II which was launched by the United States towards the planet Venus on August 27, 1962. One hundred and nine days later the spacecraft passed Venus, scanned the planet and sped on; its duty performed.

During its lifetime Mariner II did the following:

1. It performed the first and most distant trajectory-correcting maneuver in deep space, firing a rocket motor at the greatest distance from the Earth: 1,492,000 miles (September 4, 1962).

2. The spacecraft transmitted continuously for four months, sending back to the Earth some 90 million bits of information while using only 3 watts of transmitted power.

3. Useful telemetry measurements were made at another record distance from the Earth: 53.9 million miles (January 3, 1963).

4. Mariner II is the only spacecraft to operate in the immediate vicinity of another planet and return useful scientific information to Earth: approximately 21,598 miles from Venus. (December 14, 1962).

5. Measurements were made closest to the Sun: 65.3 million miles away (December 27, 1962).

6. Mariner's communication system operated for the longest continuous period in interplanetary space: 129 days (August 27, 1962 to January 3, 1963).

7. Mariner achieved the longest continuous operation of a spacecraft attitude-stabilization system in space, and at a greater distance from the Earth than any previous spacecraft: 129 days (August 27, 1962 to January 3, 1963), at 53.9 million miles from the Earth.

III. DESIGN PHILOSOPHY

The design of a spacecraft must be considered as an integrated system rather than as a coordinated assemblage of subsystems. This is especially necessary when the cost per pound of spacecraft is extremely high. Even simple planetary goals require sophisticated and advanced technology of diverse disciplines. The impulse available sets an upper bound to the weight placed in orbit. The selection of the mission is, therefore, a careful balance between the desire to extract the maximum result from what is a tremendous technological effort and the allowable weight.

It is possible to design spacecraft that are attitude free or partially stabilized and still achieve some usefulness. Examples are the Vanguard (attitude free) and the Pioneer IV (spin stabilized). Vanguard had a relatively short communication range. Pioneer IV carried its own power in the form of batteries.

In order to maintain communications independent of attitude, the antenna pattern must be omni-directional. (How does one go about achieving a truly omni-directional pattern?) Assuming that we can produce an omni-directional pattern let us proceed. Our ground-based station will be assumed to have a threshold of -156 dbm (decibels below one milliwatt) for 10 cps of data and an antenna gain of 42 db. For Venus range this requires 120 watts radiated from an omni-directional antenna to exceed this threshold.

Assuming a conversion efficiency of 10 per cent for the solar energy to electricity, a 60 per cent conversion efficiency to achieve the proper voltage levels over a large solar constant variation, and a 50 per cent over-all transmitter efficiency, we require 1.43 meters^2 of the effective surface at the Venusian range from the Sun.

This requires a spherical surface of 5.75 meters^2 or a diameter of 1.35 meters completely covered with cells. A configuration such as this assuming emissivity of .9 would heat up to a temperature of 500°F at Venus. Hardly a suitable environment. This is true if the sphere is conducting perfectly. Surely we would prefer a benevolent temperature of around 70°F.

So we desire some means of keeping the heat from the solar cells out of the instrumentation compartment. The configuration shown in Figure 1 starts to satisfy our requirements.

A slight droop of the solar panels would place the center of solar pressure behind the center of gravity and the system would lie stable towards the Sun. This would require no active attitude control if one could achieve damping. We still must radiate 120 watts omni-directionally. Solar oriented power systems weigh about two pounds per regulated watt. Obviously an antenna capable of directing the energy to earth would alleviate our weight problem. The question is whether or not this directional antenna and its associated control equipment is superior to a simpler yet heavier system.

The Mariner II attitude control system weighed 53 pounds. However, excluding power, it requires additional support from the rest of the system and has an allocatable weight of 85 pounds. The power system can be rated at about two pounds per watt of regulated power. Thus the attitude control system is the equivalent of 106 regulated or about 53 radiated watts. Figure 2 illustrates how quickly weight is saved with an attitude control system.

In addition to the initial considerable weight saving:

1. It is necessary to be looking at the planet during fly-by in order to make a direct observation.

2. It is necessary to make a turn-axis maneuver in order to connect for both miss distance and time of arrival. This requires a knowledge of the initial attitude.

3. The directionality of cosmic phenomena is desired. Therefore, the spacecraft attitude must be known.

These requirements can be met by three-axis attitude stabilization.

Physically we were limited to a 4-foot diameter antenna on Mariner II and radiated 3 watts. This provides our margin in the communication system. These numbers are not precise but should serve to illustrate the point.

I believe that it is shown that for Venus, attitude control provides tremendous benefits. The majority of the weight of the attitude control system is in equipment that is independent of the range from the Earth or time of flight. The variable dependent upon time of flight and range are the quantity of gas carried and the sensitivity of the earth seeker. By increasing the gas capacity and the earth seeker sensitivity the range can be increased.

Mr. Forney is presenting a detailed description of the Mariner II attitude control system at this meeting. I will therefore leave this subject and proceed to describe the over-all machine.

IV. SPACECRAFT SYSTEM

The spacecraft is an attitude stabilized craft using the Sun and the Earth as references. Figure 1, 2 and 3. The Sun is also used as the major source of power through the use of solar cells. A rechargeable battery furnishes power whenever the panels cannot be oriented towards the Sun. Such periods are during launch, initial Sun acquisition and midcourse maneuver.

The Earth reference is used to direct an RF beam back to the Earth in order to achieve telemetry from the planet distances.

The machine is capable of a maneuver in space if required to correct the trajectory. The magnitude of the correction is limited by the allowable fuel weight. The accuracy of the correction is limited by the stability of the gyro control system and the accuracy of the motor shutoff.

Communications is two way in the region of 960 MC utilizing phase locked techniques. The RF carrier can be used in a one-way mode, from spacecraft to Earth. However, accurate doppler can be realized by using the two-way capability. The system is so arranged that commands can be transmitted from the Earth as needed. We used eleven real time commands in the system and three stored commands.

Telemetry is digital in form. This allows what is virtually a completely automated data handling system. If required it is possible to place the data in the hands of the engineer or scientist within one-half hour after transmission from the spacecraft. A deficiency in our system, due to limited weight, is the low band width available to us that prevents the direct transmission in analog from some obviously desirable functions. For example, midcourse motor ignition and cutoff data would be of considerable interest. Table I lists the Mariner II engineering telemetry capability.

Except for the midcourse maneuver, the machine is designed to be independent of the command system. The commands are in most cases redundant with normal programmed functions. The normal functions are handled by our central computer and sequencer or CC&S. This includes such normal/abnormal conditions such as a meteoritic impact perturbing the attitude.

The midcourse motor is a monopropellant type using anhydrous hydrazine. A system of regulated high pressure nitrogen supplies a near constant thrust. At present the system is designed for a single operation.

The scientific instrument package is grossly divided into two areas of interest. One is the planetary data and the other the interplanetary data. Most of the interplanetary data is of interest in the vicinity of the planet, also. However, the microwave and infrared radiometer are not exercised, except in a calibrate sequence, other than during planetary encounter.

The greatest consumer of power is the communications system. Even with our ability to detect signals of the level of 10^{-19} watts, the question of RF energy on board the spacecraft and its attendant antenna system primarily govern not only the weight of the spacecraft but also its configuration.

The spacecraft configuration must take into account the look or angular accessibility external to the spacecraft. It is unfortunate that a device that will spend its functional lifetime in a gravity free and unconfined environment must be designed to withstand the confinement and pummeling imposed by the launch vehicle.

The major competitors for look angle are the solar panels, the high-gain antenna, the scientific instruments, the midcourse rocket motor, and the so-called omni-antenna. The first four are the obvious. The last is a result of our present design philosophy. This "omni" pattern allows us to receive telemetry prior to attitude acquisition. Also part of this omni-directional coverage is our desire to be able to send commands to the spacecraft independent of the attitude.

On Mariner II "omni" system this consisted of two antenna systems, one for telemetry and a set of receiving antennae.

The solar panels on Mariner R were 27 feet^2 and contained 9,800 solar cells. They were sized to supply the power a minimum of 148 watts at Earth. The power estimates were modified after the design was well under way and this panel had been constructed. Later data on susceptibility of N on P solar cells to radiation raised the concern that a major solar flare could degrade performance to an intolerable level. The additional tip as seen in Figure 3 was added to increase our confidence that we would have sufficient power at the planet. The other panel was tipped with a section of rubberized dacron so that the solar pressure would be balanced.

The scientific accomplishments of the Mariner II have been extensively reported. The engineering aspect has received lesser attention by the public. Since this paper is presented to what is primarily an engineering conference, I will concentrate on the engineering results and refer the reader to the available literature for the scientific aspects.

I will quickly review the entire flight history and then summarize the performance of the individual subsystems.

V. FLIGHT HISTORY

Launch of the Mariner R spacecraft atop an Atlas Agena vehicle took place on August 27, 1962, 6 hours 26 minutes and 13.927 seconds GMT, 26 minutes and 3.08 seconds later injection occurred. This initiated the most successful planetary spacecraft flight thus far by the United States.

Upon separation all events required were commanded and performed in the predicted manner. Data from the gyros indicated a tumbling rate upon separation of plus 675° per hour in pitch and minus 400° per hour in roll. Yaw was saturated and, therefore, can only be noted as exceeding 1800° per hour. Acquisition of the Sun was achieved within five minutes after the CC&S issued the command.

Some difficulty was experienced in acquiring RF lock from the South African station during this period of tumbling. The difficulty apparently cleared when tumbling ceased.

Upon Sun acquisition the power available exceeded our initial prediction with a surplus of about 43 watts.

A significant abnormality was noted at this time. The indication was that the earth sensor temperature was high by about 30°F.

On the 29th of August the command to turn on cruise science was sent and the bit rate was changed from 33 to 8-1/3 bits per second and remained at this rate for the remainder of the mission as planned.

All scientific instruments operated satisfactorily. Of considerable interest was the calibration in space of the magnetometer. Prior to Earth acquisition the spacecraft is free to roll. This allowed an accurate method of subtracting the spacecraft perm or magnetic field in the X and Y axis.

The attitude control system at this time was using about .025 pounds of gas per day. The roll rate was at 1.6 degrees per minute; this roll rate rose to 11.2 degrees per minute by the time of Earth acquisition. It is believed that this was due to the naturally small leakage in the roll valves. However, it was well within the specification.

Earth acquisition was performed automatically by CC&S command at about 0530 GMT on September 3, 1962. All events were normal except that the system switched to the high-gain antenna immediately instead of waiting until Earth acquisition. Based upon the roll rate and the time it took for acquisition to actually occur it appears quite probable that the Earth was imaged on the sensor when turn on occurred. This switched the high-gain antenna on, but the roll rate was too high and the system overshot the target. It then rolled about at the rate set by the gyro system and acquired the Earth the next time around. In the meantime, several dozen cardiac cases developed as the signal level from the spacecraft dropped as the antenna beam swung away from the Earth. You could track the signal level return by the change in complexion of the observers.

An abnormality was then noted. The earth sensor telemetry indicated a light intensity much lower than predicted. It was suspected that the sensor had locked onto the Moon. However, the antenna hinge angle indicated properly for the Earth-Probe-Sun angle. Thus the dilemma was presented to our Space Flight Operations:

1. Was the sensor locked onto the Moon thereby giving us an incorrect initial condition for the midcourse maneuver?

2. Was the earth sensor bad?

3. Was the telemetry incorrect?

4. Was the antenna hinge angle sensor stuck?

Because of this dilemma the midcourse maneuver was delayed one day. On the next day by observing the normal performance of the high-gain antenna and the Moon-Probe-Sun angle it was agreed that the earth lock was correct and that an abnormality existed in the earth sensor.

On September 4, 1962, the spacecraft midcourse maneuver was commanded to correct the 400,000 km planetary miss. The spacecraft was commanded to roll 9.33 degrees and then pitch 139.85 degrees and then to insert a velocity of 31.16 meters per second. At 3:49 P.M. the execute command was sent. The sequence then proceeded as expected; reacquired the Sun and then the Earth. Again the high-gain antenna was automatically switched prematurely.

The midcourse performance was high in velocity by about 4.5 per cent with about a 3 degree pointing error. The correction was quite satisfactory as illustrated in Figure 4.

There was heat transient greater than predicted and the battery case heated to its maximum upper limit of 120°F. The thermal control louvers opened to their maximum. It is certainly fortunate a longer burn time was not required for the maneuver.

On September 8, 1962 at about 1250 GMT, there was a momentary loss of Sun. This is believed due to a small meteoritic impact. The spacecraft quickly recovered and proceeded in its normal cruise mode.

Normal except for the very unhappy earth sensor situation. Based upon the telemetry data it was predicted that earth lock would be lost between the 10th and 15th of October.

An emergency plan of action was evolved that <u>might</u> have salvaged the mission.

I say 'might have,' because on September 29, 1962 what might have been a meteoritic impact occurred. The spacecraft initiated its recovery sequency and when the cosmic dust had cleared the earth seeker intensity telemetry indicated the proper value. (Figure 6)

As any housewife knows a few bangs will make any electronic device work properly.

On October 31 telemetry indicated a problem in the power system. The diagnosis indicated a short circuit in one of the solar panels. This short circuit drew sufficient current to cause concern that the power available would be exceeded and require support from the battery. A command was sent from our Goldstone Tracking Station to turn off the scientific instruments.

Eight days later the short circuit cleared itself and the scientific instruments were commanded to turn on again. On November 15 the trouble appeared again. However, by this time the solar constant had increased due to the closer proximity to the Sun and one panel was sufficient to supply all the power needed by the spacecraft.

On December 9 we lost four telemetry measurements. These were the high-gain antenna hinge position, the midcourse motor propellant tank pressure, the midcourse motor pressure, and the attitude control nitrogen pressure. This was diagnosed as a blown fuse in our data encoder. The midcourse motor data did not disturb us since the maneuver had already been performed. The attitude gas supply at last measurement indicated a reserve sufficient for 100 days.

On December 12, just two days prior to encounter, the central computer and sequencer failed to issue the 155th and all following cyclic pulses. The CC&S was designed, among other functions to provide a timing pulse or cyclic update every 1000 minutes to the attitude control subsystem to update the antenna reference hinge angle. This reference is used by the high-gain antenna if the earth sensor should lose lock.

On December 14 the CC&S was also programmed to switch on the encounter sequence twelve hours prior to encounter in order to allow sufficient time for calibration of the planetary scientific instruments. This event did not occur and was attributed to the same failure noted two days earlier.

Forewarned, the backup command was ready and transmitted. The spacecraft responded properly. Engineering telemetry was turned off and the planetary experiments, microwave radiometer, and infrared radiometer were turned on.

The scan performed well. The pattern achieved is shown in Figure 5.

The spacecraft signal was received until January 3, 1963. At 0512 GMT the subcarrier data was lost. At 0700 GMT the carrier was lost and the mission was considered as terminated.

It is our belief that the earth seeker sensitivity was exceeded and the gyros turned on in a vain attempt to reacquire. This was maintained until the gas supply was depleted. The Mariner II is now a space derelict.

VI. ENGINEERING RESULTS

The more interesting engineering results are presented by subsystems.

A. Structure

The primary purpose of a spacecraft structure is to support all subsystems in an optimum configuration. In addition, the spacecraft structure must be designed to be compatible and function properly with the booster. This includes separation, shroud clearance, and environment survival.

The configuration appears to have been adequate. The spacecraft over-all performance seems to indicate no structural failures.

B. Temperature Control

Prior to launch the environment was maintained within the shroud at 70°F. The spacecraft temperature at launch was 109°F. Two hours after launch the temperatures dropped slowly. Eight hours after launch the temperatures had stabilized within the hexagonal instrument compartment to an average of 84°F.

Temperatures remained essentially constant from this time until midcourse maneuver. At this time, due to increased internal power, a significant heat input from the propulsion system, and a loss of Sun orientation required by the maneuver, the spacecraft hex experienced a 20°F average rise. Within ten hours after the maneuver, the temperatures returned to their pre-midcourse maneuver values. The maximum and minimum temperatures measured during the maneuver were 130°F on the motor nitrogen tank and 72°F on the upper thermal shield.

Temperature then increased steadily until October 31 when the science instruments were commanded off to accommodate the solar panel problem. Temperatures dropped about 5°F when the power dissipated in the compartment decreased by 12.5 watts.

? Eight hours after cruise, science was reactivated and the temperatures rose to their previous values.

Temperatures were not telemetered during the encounter mode, but temperatures measured before and after encounter were compared to determine the thermal influence of Venus on the spacecraft. Both the battery and the power assembly indicated a 2°F rise when the cruise mode was resumed. Both of these assemblies faced Venus during encounter, but part of the temperature rise resulted from increased internal power.

In general, the temperatures were higher than predicted. Temperatures near Earth exceeded expectations by as much as 40°F; those near encounter were as much as 75°F higher. The only monitored temperatures that behaved as expected were the solar panels. The predicted and measured temperatures are shown in Figure 6.

The probable sources of error are as follows:

(a) Greater than expected power dissipation although there does not appear to be a great disparity between expected and actual power dissipation.

(b) The fact that the temperature rise between Earth and Venus was substantially higher than expected, suggests that excessive solar inputs were partially to blame. These inputs could have been caused by one or all of three of the following: (1) The energy reflected from the various structural members was greater than anticipated, (2) conduction from structural members was greater than expected, and (3) degradation of the white paint and upper thermal shield because of ultraviolet irradiation may have caused an increase of solar absorbability.

(c) The emissivity may have been lowered by contamination of the surface.

(d) Errors in testing which were performed with heating pads. This could have caused local "hot spots" which would radiate more heat than expected.

In general the design was adequate.

The louvers performed well, decreasing the average compartment excursion by 12°-15°F.

C. Central Computer and Sequencer:

Computation for the subsystems issuance of commands, and the basic timing is supplied by the CC&S.

Its performance during launch and initial cruise was flawless. The midcourse maneuver had an error that might possibly have been caused by not all of the pulses from the digital accelerometer being counted as it measured the velocity increment imparted by the midcourse rocket motor.

There was an error in the CC&S frequency that is correlated to the change in temperature.

As was noted previously the cyclic failed two days prior to encounter. The failure appeared to be in the frequency counter chain and was probably due to the high temperatures.

D. Attitude Control

This area is covered in detail by Mr. Forney's paper, "Mariner II Attitude Control System," also presented at this conference. Therefore, I will not dwell on this subject.

E. Power

The function of the power system is to provide a central supply of electrical energy to operate the equipment on board the spacecraft. The battery supplies power for the periods during which the attitude towards the Sun is deliberately or accidentally disturbed.

Performance of the power system was good except for those events previously noted where the failures can possibly be correlated with some external influence.

The attached Figure 7 shows the drain upon the battery as a function of time. Since the battery provides power when the panel capacity is exceeded, it clearly shows that it was not called upon for support after the event of September 8. Figure 8 shows the power available to the spacecraft during encounter and the actual load. The margin of safety was quite adequate even if one assumes a 20 per cent degradation due to solar proton activity.

F. Propulsion Subsystem

The function of the Mariner II midcourse propulsion subsystem is to reduce the dispersion at injection due to the launch vehicle.

Prior to the required maneuver, the propellant tank pressure maintained a rising characteristic which had been noted prior to launch. This appears to have been due to hydrazine decomposition resulting from incompatibility with the expulsion bladder containing the propellant. This resulted in a non-standard start transient. However, no difficulties were realized as a result of this transient.

For the midcourse maneuver a velocity increment of 31.16 meters per second was required and commanded. With a spacecraft of 447.67 pounds this is a predicted burn time of 28.3 seconds (this includes starting sequence). The doppler shift data roughly verified this prediction.

An estimate of the velocity increment based upon nitrogen and propellant tank pressures and temperatures was calculated. Based upon this telemetry data, a velocity increment of 37.5 meters per second was calculated. This 20 per cent discrepancy is well within the accuracy of the computation and does not necessarily indicate an error in the execution of the maneuver.

G. [illegible] Subsystem

The purpose of the spacecraft radio frequency system is to coherently receive a phase modulated signal transmitted from an Earth-located transmitter and to transmit back to Earth a phase modulated signal which is either coherent with the received signal or derived from a crystal controlled oscillator.

The RF subsystem operated normally through the entire flight. Telemetry indicated a drop of transmitted power of about 1 db and a drop of the automatic gain voltage of about 0.2 volts d.c. Both of these are in agreement with a temperature increase of from 85°F near Earth to 152°F at Venus encounter.

A comparison of the actual and theoretical received signal shows a good agreement well within the tolerance. Figure 9 shows the signal strength calculated for the Woomera Station and the measured result. It also shows the calculated signal for the spacecraft receiver and the telemetered value.

The command system performed flawlessly and provided functional redundancy. As has been noted previously these were indispensable in the success of Mariner II.

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TABLE I

MEASUREMENT

1. Battery Voltage
2. Yaw Control Gyro
3. Pitch Control Gyro
4. Roll Control Gyro
5. Battery Current Drain
6. Pitch Position Sun Sensor
7. Yaw Position Sun Sensor
8. Roll Error Earth Sensor
9. Event Registers
10. Command Detector Monitor
11. Earth Brightness
12. Antenna Ref. Hinge Angle
13. Antenna Hinge Position
14. Transponder AGC (Coarse)
15. Receiver Phase Error (Coarse)
16. Propellant Tank Pressure
17. Battery Charge Current
18. M/C Motor Nitrogen Pressure
19. Receiver Phase Error (Fine)
20. Directional Antenna RF Power
21. Louver Position
22. 4A11 Solar Panel Voltage
23. Omni Antenna RF Power
24. A/C Nitrogen Pressure
25. 4A11 Solar Panel Current
26. 4A12 Solar Panel Voltage
27. 4A12 Solar Panel Current
28. Power Boost-Reg. Temperature
29. M/C Nitrogen Tank Temperature
30. Propellant Tank Temperature
31. Earth Sensor Temperature
32. Battery Temperature
33. A/C Nitrogen Temperature
34. 4A11 Panel Temperature (front)
35. 4A12 Panel Temperature (front)
36. 4A11 Panel Temperature (back)
37. Case I Temperature
38. Case II Temperature
39. Case III Temperature
40. Case IV Temperature
41. Case V Temperature
42. Lower Thermal Shield Temperature
43. Upper Thermal Shield Temperature
44. Plasma Experiment Temperature
45. Antenna [illegible] Temperature

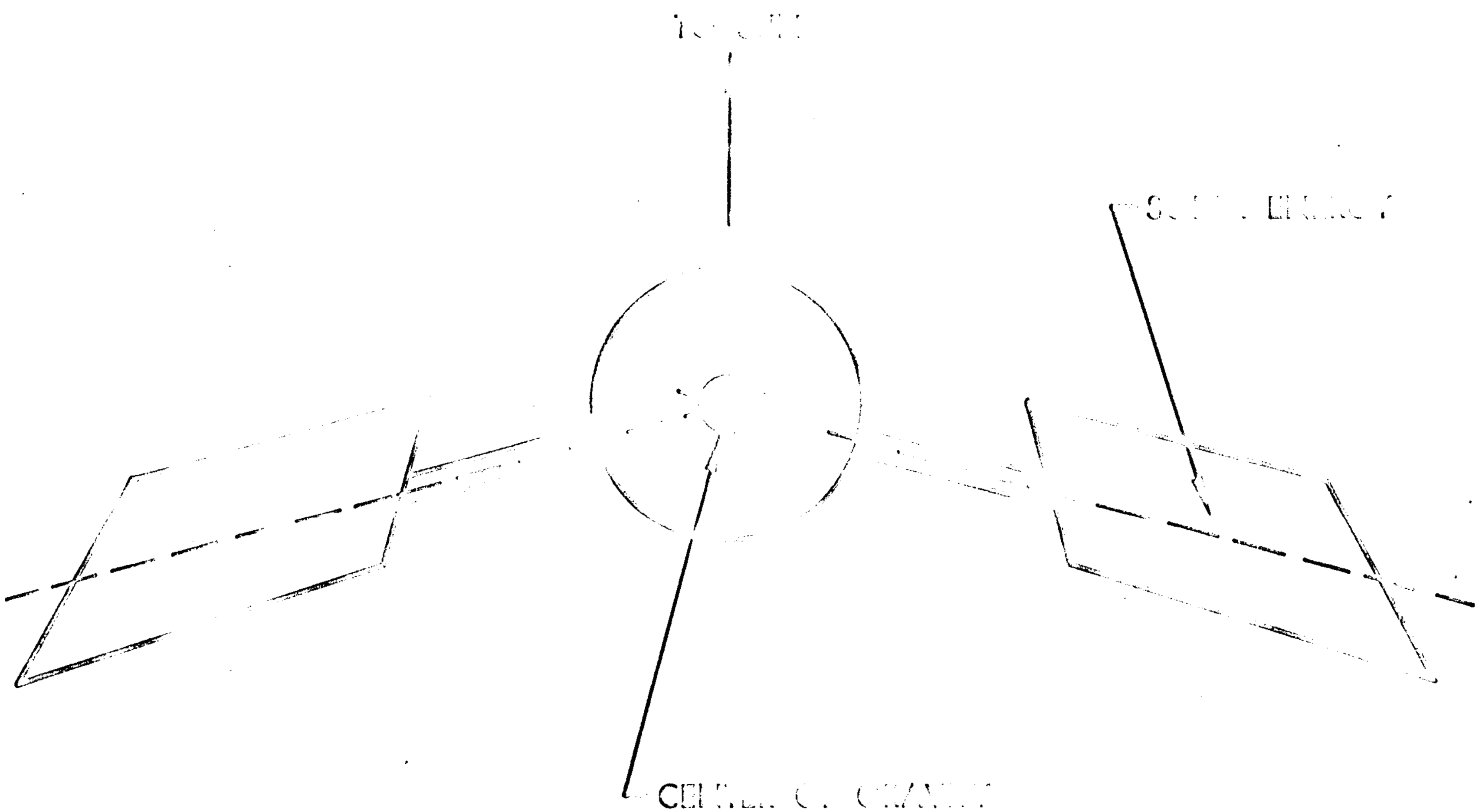


FIGURE 1. SOLAR STABILIZED CONFIGURATION

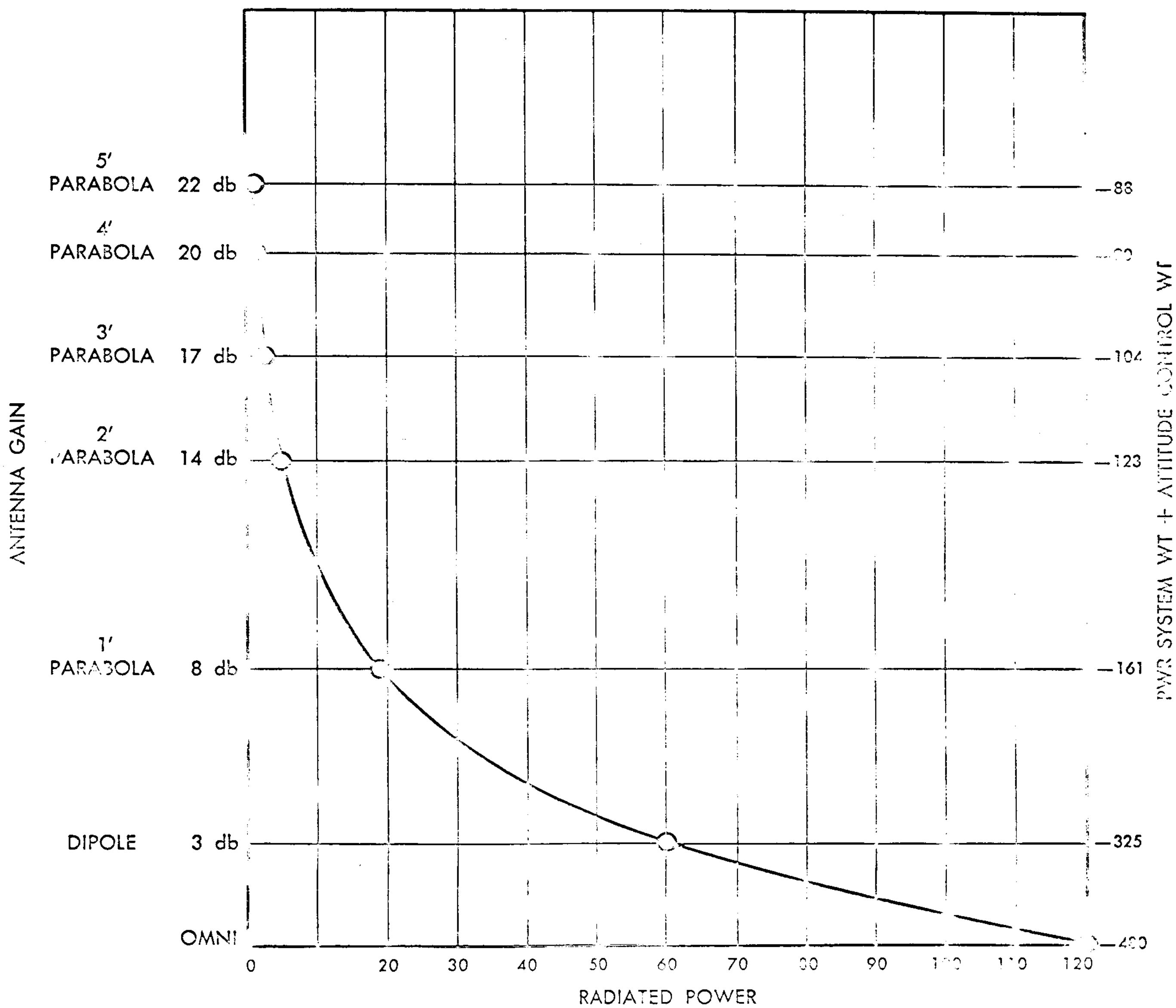
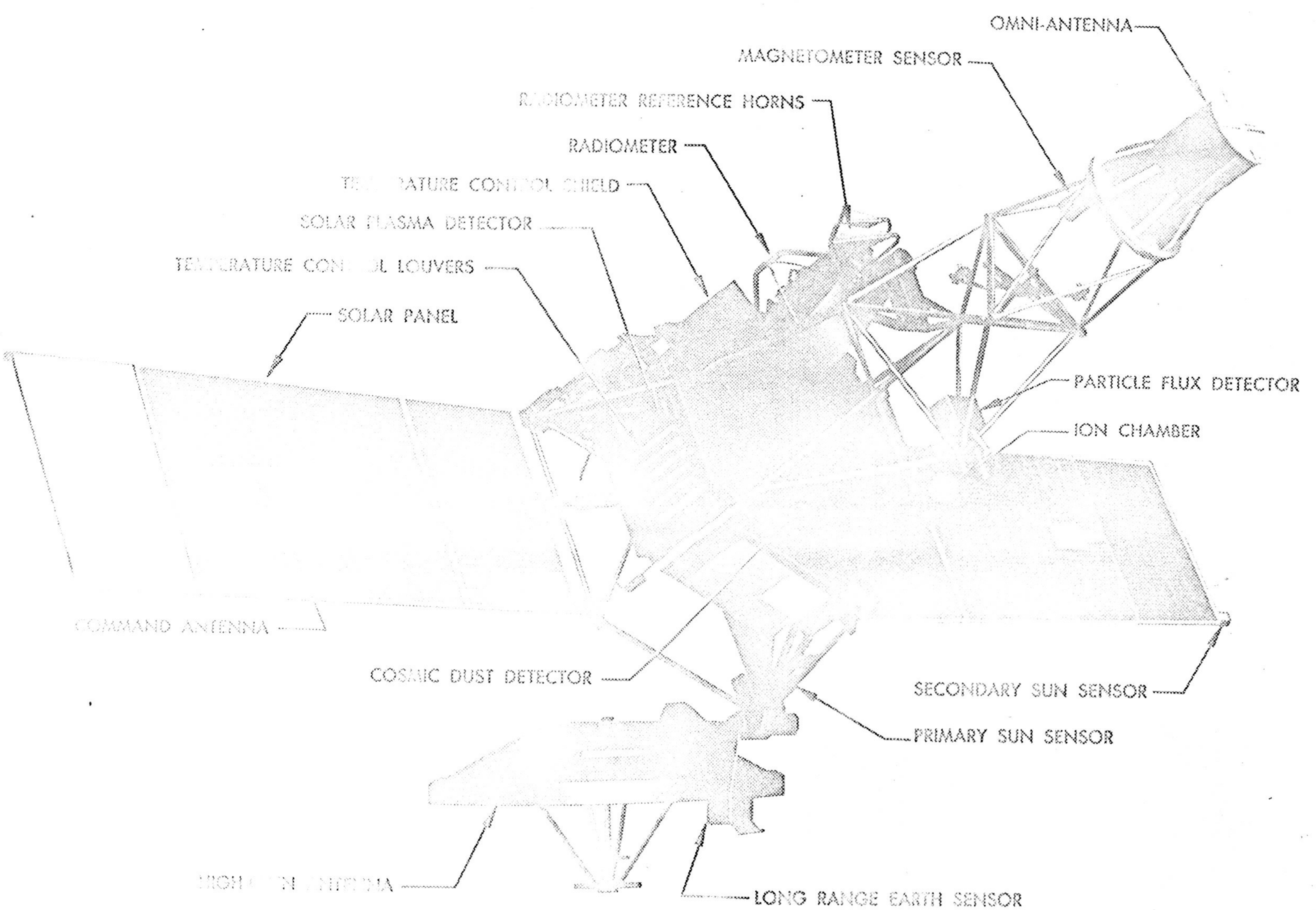


FIGURE 2. WEIGHT SAVING ACHIEVED BY ATTITUDE CONTROL



R II SPACECRAFT

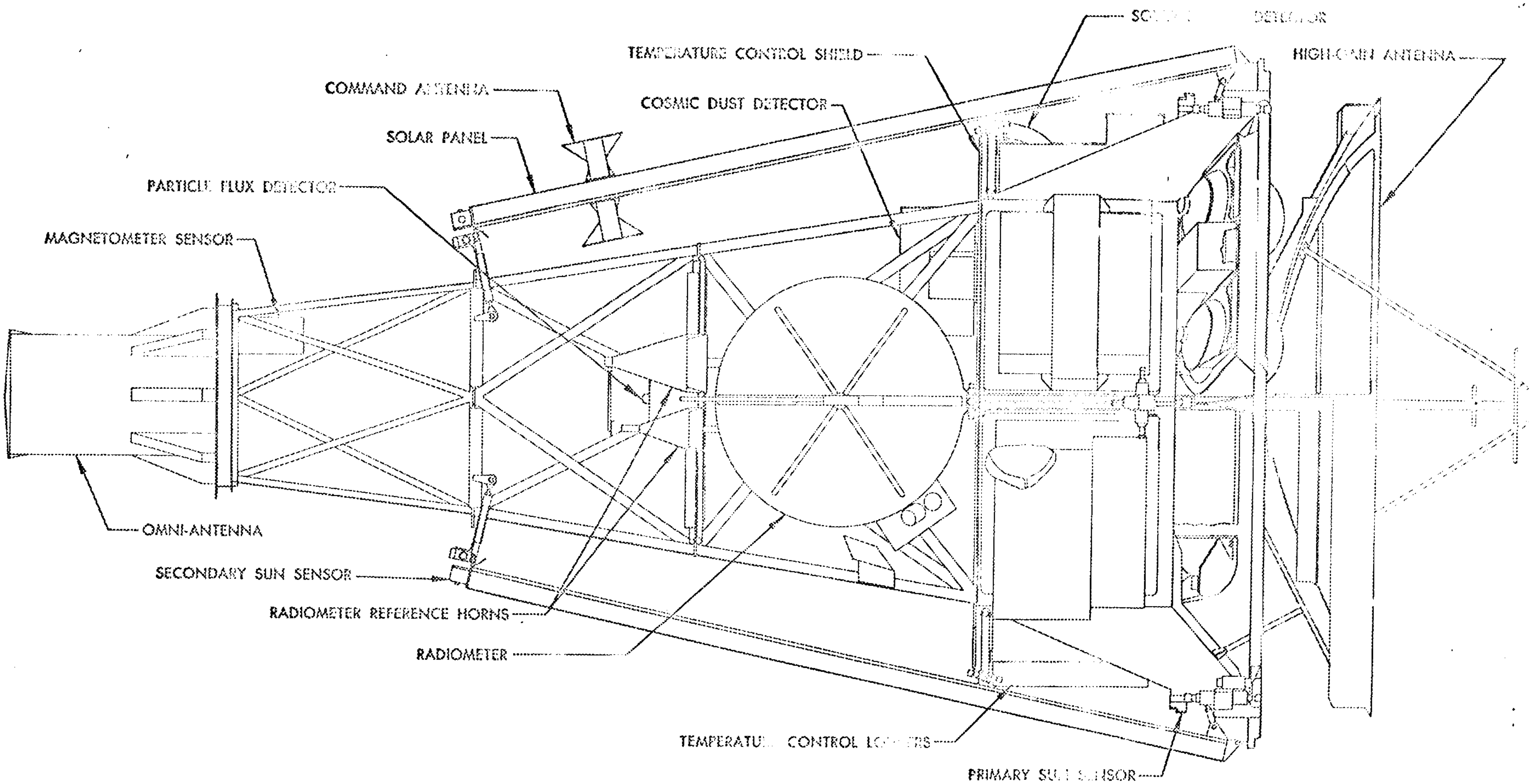


FIGURE 3B. MARINER II SPACECRAFT

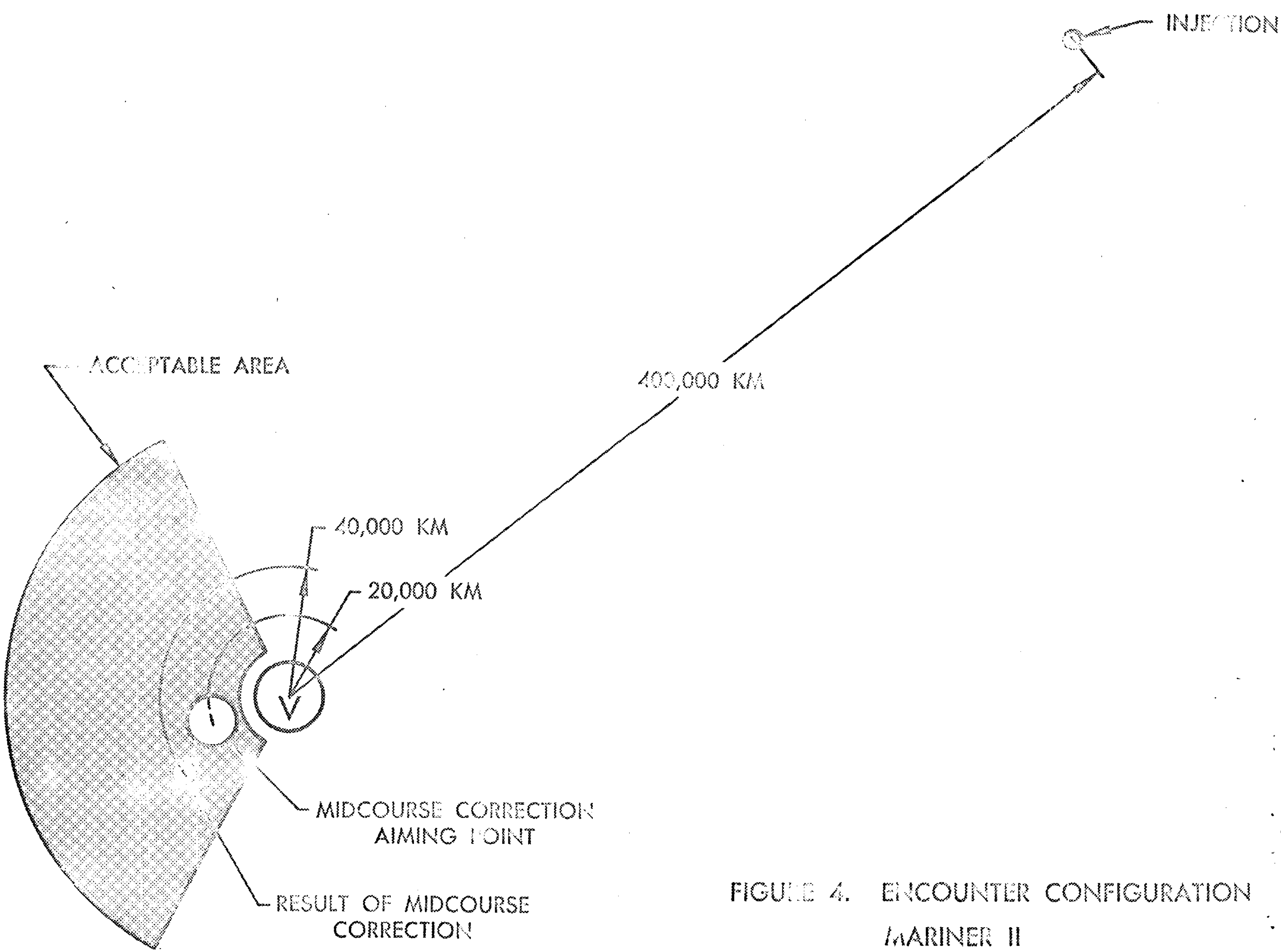


FIGURE 4. ENCOUNTER CONFIGURATION
MARINER II

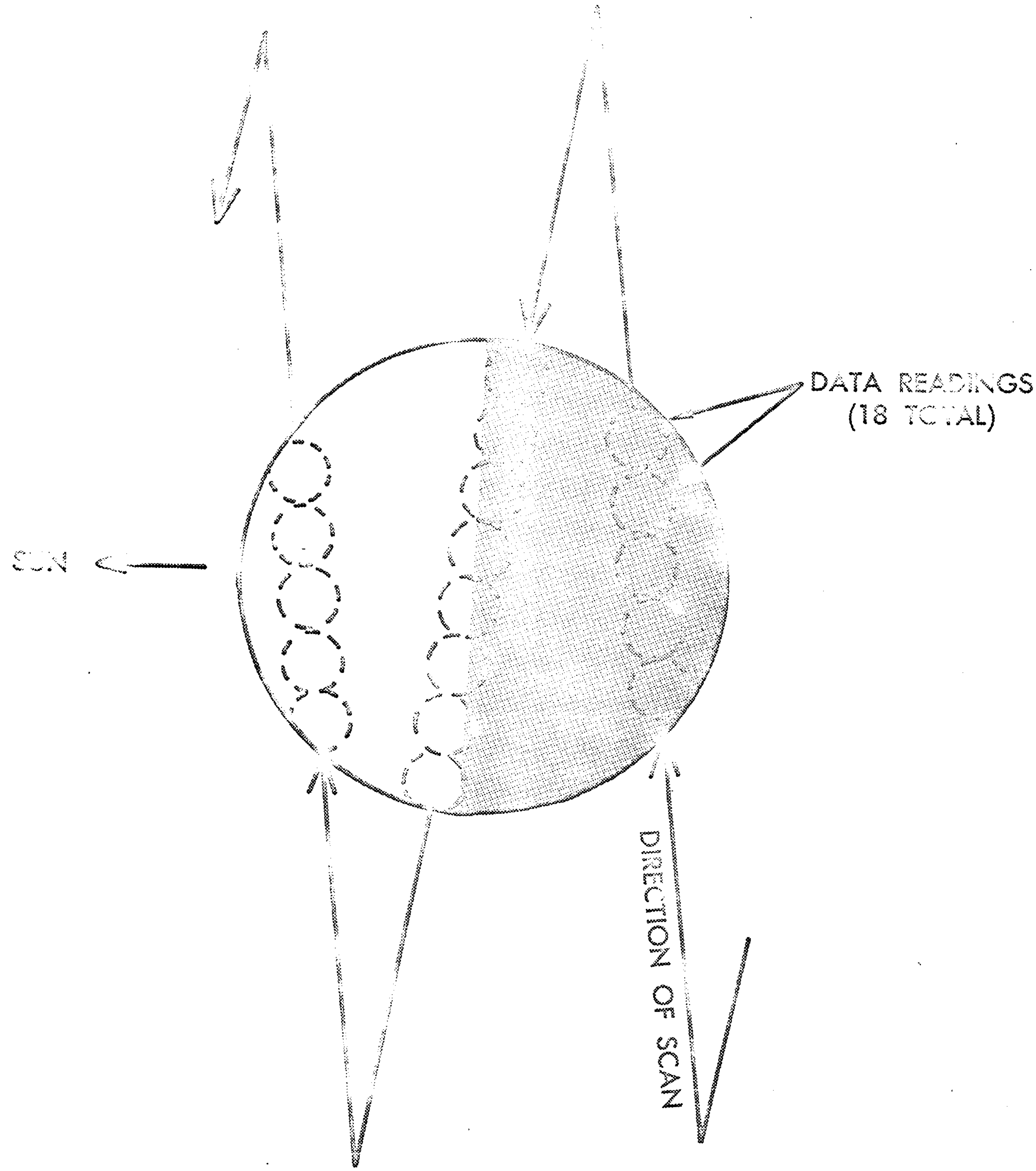


FIGURE 5. MARINER II RADIOMETER SCANS OF VENUS

Component	Earth (stabilized) (°F)		Venus (°F)		Maximum (Jan. 2, 1963)	Desired Operating Limits (°F)
	actual	predicted	actual	predicted		
Pwr boost reg.	80	78	129	114	143	32-140
M/C nitrogen	78	55	139	84	151	35-165
Prop. tank	76	55	138	84	148	35-165
Earth sensor	78	40	165*	90	171*	0-95
Battery	70	55	130*	91	141	50-120
A/C nitrogen	68	59	160*	115	-	32-140
Solar panel front	126	132	250-254*	262	-	as cold as possible
Case I	73	50	152	92	160	14-149
Case II	85	60	152	90	159	0-140
Case III	86	62	149	89	157	0-149
Case IV	74	60	124	80	134	50-130
Case V	86	52	135	84	158	32-140
Lower thermal shield	58	32	122*	58	-	-
Upper thermal shield	59	80	153	215	162	-
Plasma Expt. (Case I)	78	50	155	92	-	14-158

*Extrapolated

FIGURE 6. PREDICTED AND FLIGHT TEMPERATURES

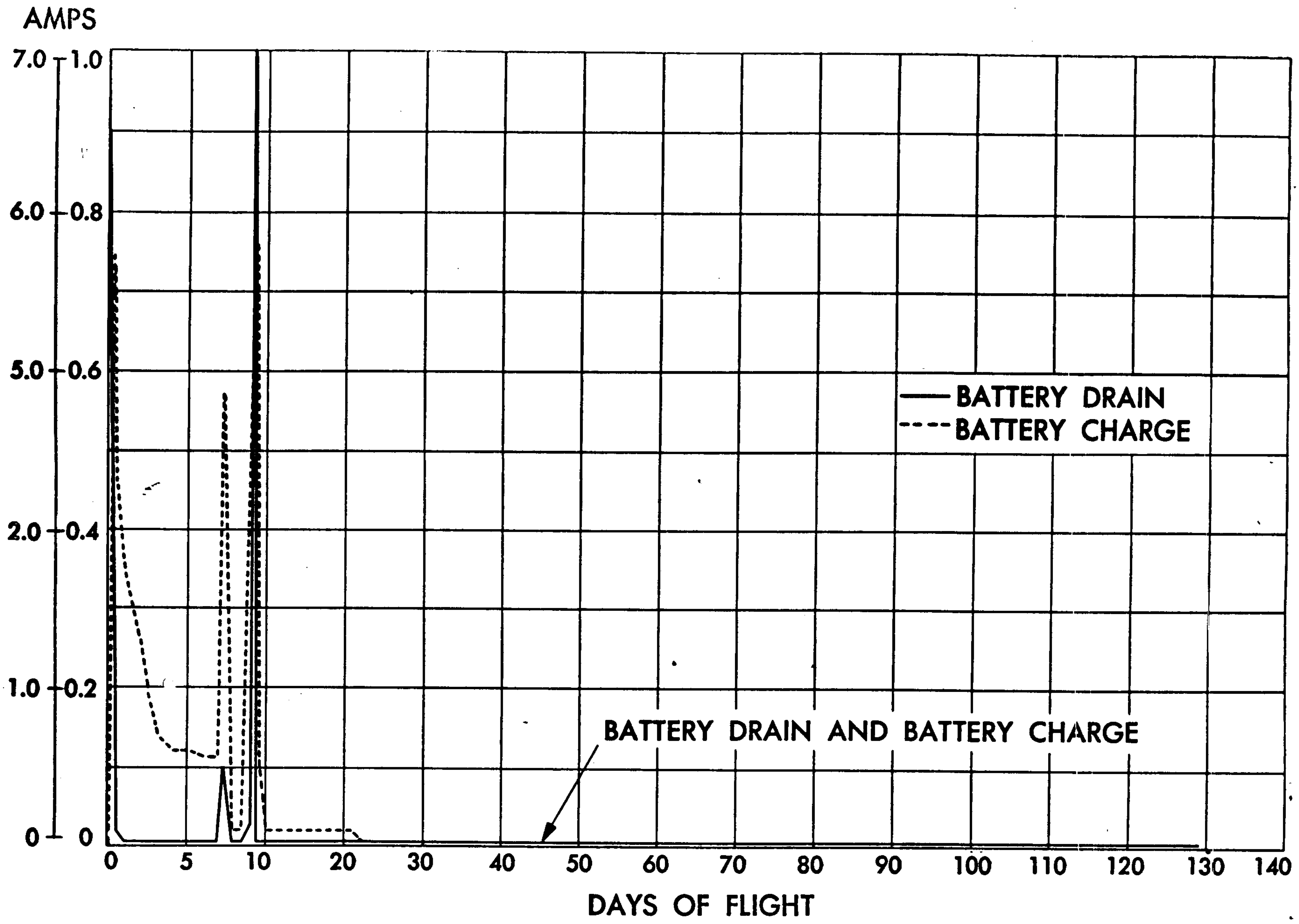


FIGURE 7. BATTERY DRAIN AND BATTERY CHARGE VS TIME

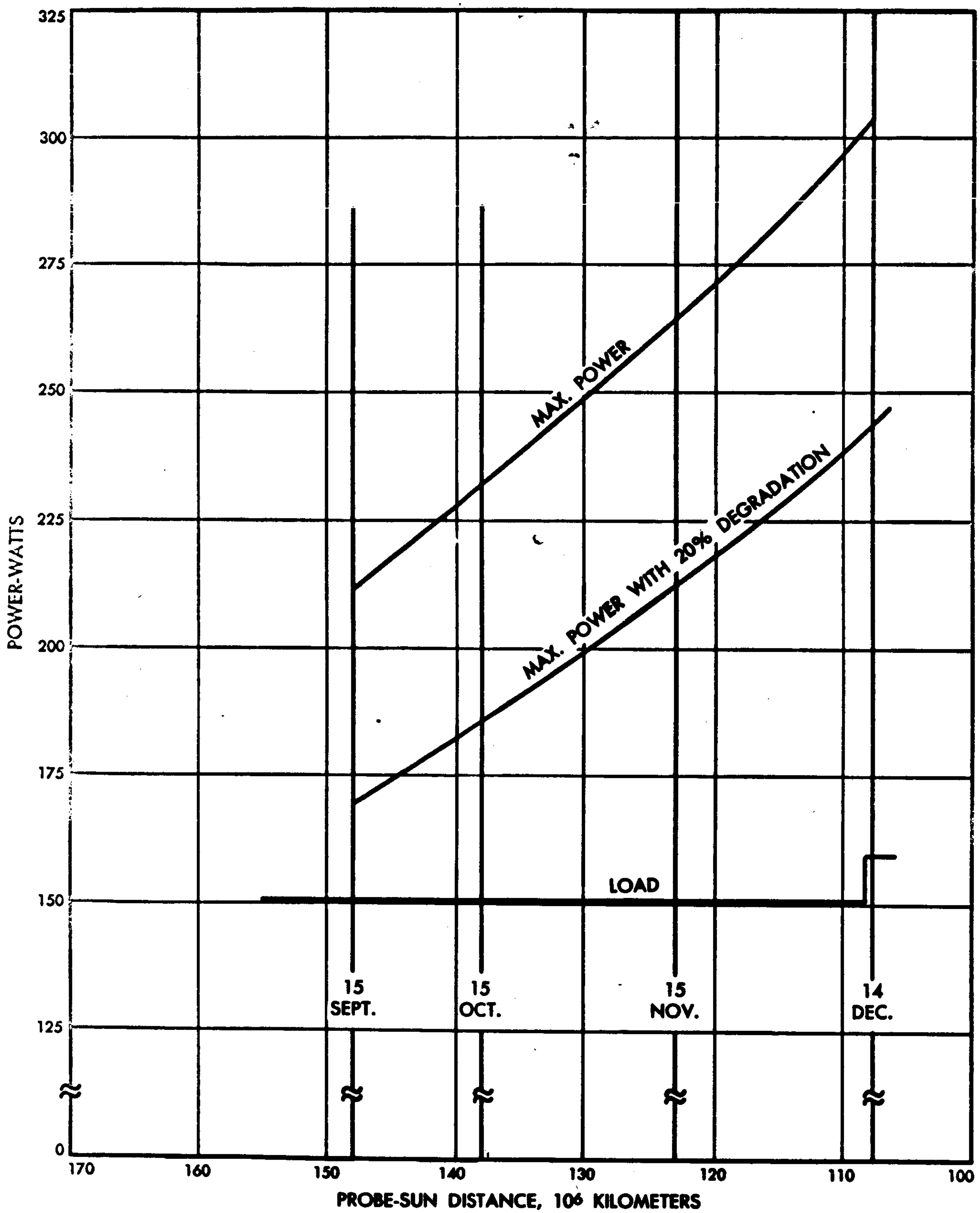
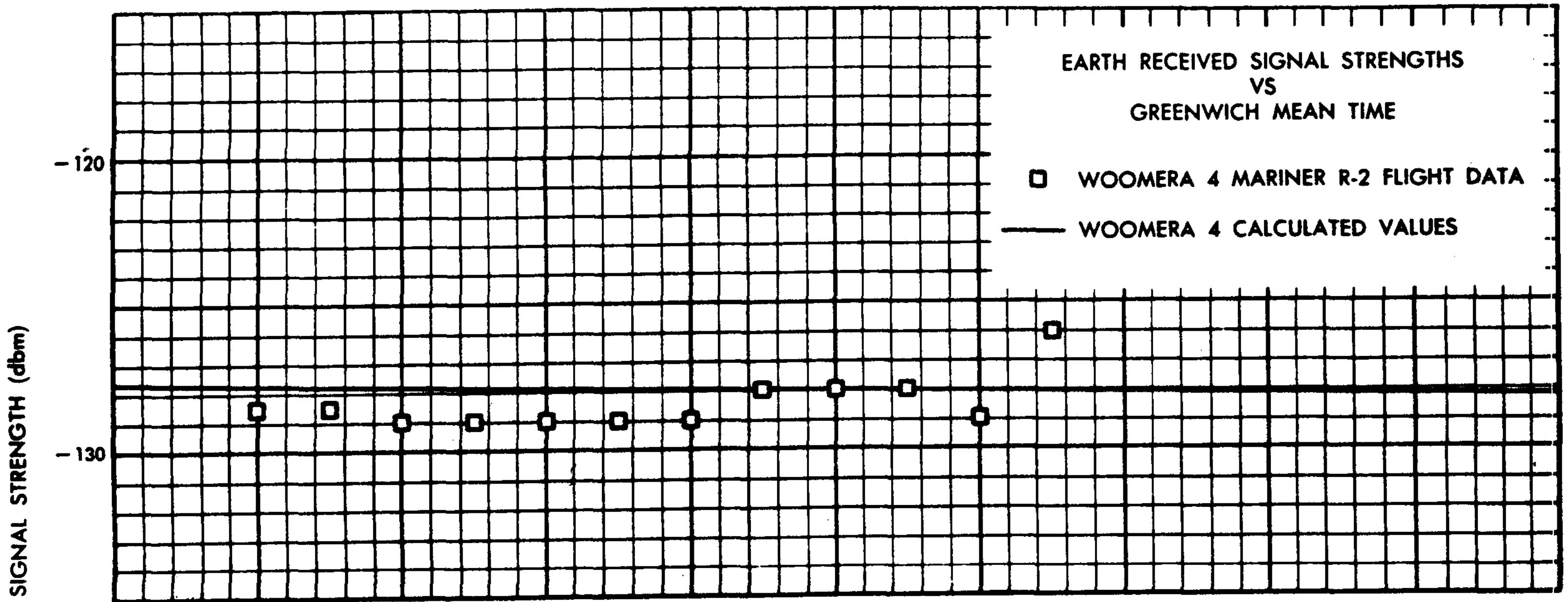
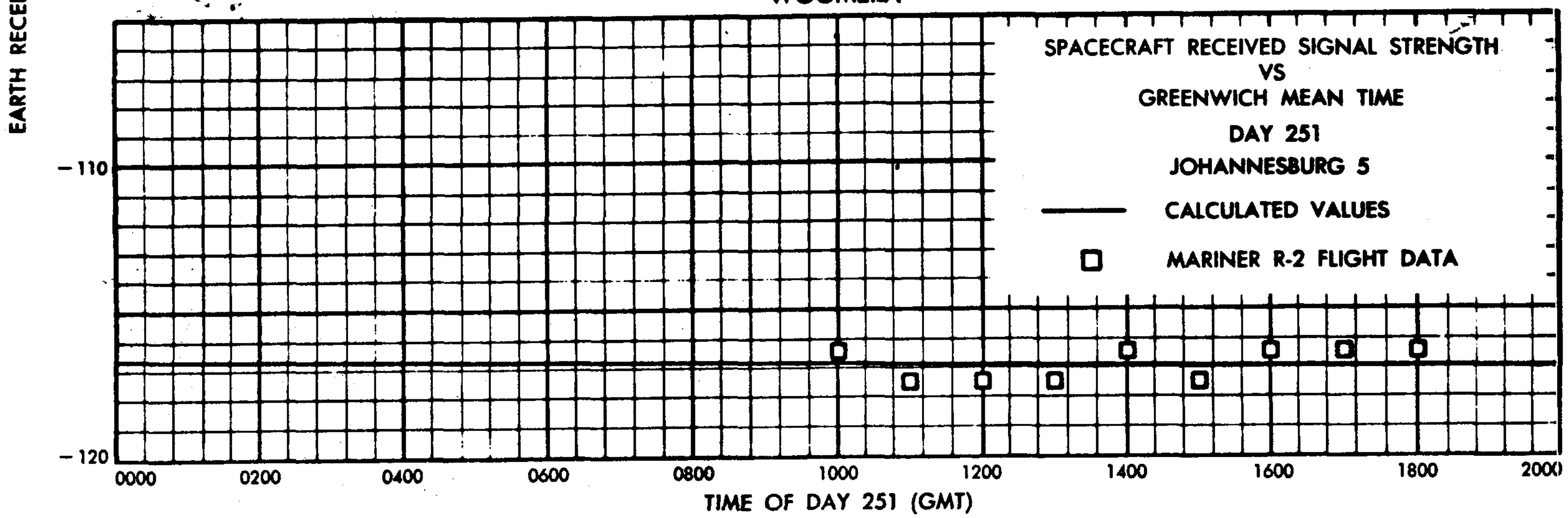


FIGURE 8. MARINER II AVAILABLE POWER



TYPICAL EARTH-RECEIVED SIGNAL
WOOMERA



TYPICAL SPACECRAFT RECEIVED SIGNAL
JOHANNESBURG

FIGURE 9.